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Current and Future Technology in Radial and Axial Gas Turbines

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CURRENT AND FUTURE TECHNOLOGY

IN RADIAL AND AXIAL GAS TURBINES

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INTRODUCTION

Knowledge and technology in gas turbines improved rapidly in the decade following World War II. During that period much of the ground work in two-dimensional flow analysis and turbine cooling was done. Convection, film, and transpiration cooled vanes and blades were studied analytically and experimentally. A cooled turbine with aluminum blades was run with a 1650 K (2500° F) inlet temperature in 1947. Also, an engine test program utilizing liquid hydrogen for a turbine cooling air heat sink ran a successful test with a 3000° F turbine inlet temperature. While there were severe operating problems that precluded practical use of these concepts at that time, these experiments did demonstrate the potential of turbine cooling. Since that time, turbine inlet temperatures in production engines for civil aircraft have increased from 1000 to 1650 K (1400° to 2500° F), largely through experience, developing technology, improved materials, and more rigorous design techniques. Figure 1 shows the history of this temperature increase from the early turbojets to the current high bypass turbofans. Note the leveling off in recent years. This will be discussed later.

Currently, quasi-three-dimensional inviscid flow analyses are standard design tools for flow analysis and these are used with empirical and calculated boundary layer and endwall effects. Hot gas side heat transfer calculation methods range from the use of flat plate heat transfer correlations to fairly sophisticated boundary layer codes with various turbulence models, transition criteria, and geometric provisions. Internal flow and heat transfer calculations for coolant side heat transfer typically use one-dimensional convection models with empirical correlations for impingement, pin fins, and turbulence promoters. Metal temperature prediction systems use these heat transfer calculations with two-dimensional conduction codes that employ paired rows of elements, inside and outside, along the blade profile at several spanwise positions. The calculated temperatures are then tied together spanwise, recalculated if necessary to reflect spanwise conduction, and used to evaluate the cooling system. This method provides an approximate three-dimensional solution for metal temperature distribution.

While currently used design methods provide turbines with high aerodynamic efficiencies, substantial uncertainties continue in the prediction of local metal temperatures. Figure 2 shows the learning curve in turbine design technology as I see it and also shows an assessment of the current ability to predict metal temperatures. Note that gas-side heat transfer coefficient uncertainty is 35 percent while the coolant side uncertainty is 25 percent. These uncertainties, combined with only approximate knowledge of gas and coolant temperatures, lead to an uncertainty of about 100 K (180° F) in local

metal temperature. This in turn leads to uncertainty in life prediction of a factor of ten. An analysis of this subject was presented at the 1980 Joint ASME/AIChE National Heat Transfer Conference, reference 1. It should be noted that a significant contributor to this limitation on metal temperature prediction ability is the lack of precise knowledge of the real engine environment. There is a real need for accurate measurements of temperature, pressure, and turbulence distributions. The price paid for this uncertainty in metal temperature prediction can be quite high in terms of development cost as well as subsequent maintenance costs. Figure 3 shows a pie chart on the cost of development of a new engine. About four years ago I surveyed four or five of the major engine companies to get this information. This representation averages the response. The total engine development costs in 1979 ranged from $500 million to $1.2 billion. In 1983 this would scale up to about $600 million to $1.6 billion. Of that total cost, 10 to 40 percent were incurred in the core turbine. And two thirds of that was in fixes. That amounts to about $40 to $400 million in core turbine changes through flight certification and perhaps one year of operating experience. Recent conversations with, and research proposals from, the engine companies indicate that this picture has not significantly changed.

Current efforts in computational fluid mechanics, instrumentation, and computer technology hold promise of another period of rapid advancement in turbine design technology. We won't see dramatic increases in turbine inlet temperature such as the 550 K (1000° F) increase in the 50's and 60's, but we will see greatly increased computer involvement in design optimization and the simulation of component and full engine operation. For the aerodynamic and heat transfer codes, we will see significantly more accurate definition of boundary conditions because of improved high temperature instrumentation and greatly improved modeling of combustors. Figure 4 summarizes very simply the changes we can expect in the next ten years. I believe that three-dimensional viscous computer codes will be standard design tools, and that the uncertainty margins in heat transfer coefficients will be reduced by a factor of three. This, in combination with accurate knowledge of the environment, should permit metal temperature prediction with an accuracy of 14 to 28 K (25° to 50° F) and greatly reduced component and engine development times and costs. We can also expect major improvements in specific fuel consumption, thrust-to-weight ratios, and time between engine overhauls.

A great deal of interesting work is underway in research, development, and design technology. High fuel costs have resulted in increased emphasis on high cycle efficiency and consequently higher cycle pressure ratios and turbine inlet temperatures. Much current work, therefore, is in high temperature turbines with complex cooling systems that employ rather hot cooling air. Many innovative aerodynamic concepts are also being explored. These include leaned or bowed vanes with contoured endwalls, variations in radial distribution of work, winglets, various endwall trenching and grooving, and several variable geometry techniques. The programs to be highlighted herein are in radial and axial turbines for the 1990's and beyond. Also, mission and cycle studies for aircraft of the period will be discussed along with the long range turbine performance goals.

RADIAL TURBINES

General Characteristics

Radial turbines are suitable for many applications ranging from very small superchargers to hydroelectric power plants. In all of these applications they show many desirable characteristics such as high efficiency, ease of manufacture, sturdy construction, and high reliability. Francis and Kaplan turbines are employed in hydroelectric systems generating more than 100 000 KW. These typically have variable radial-inflow stator vanes. The Francis has a radial inflow rotor with blades extending partially around the bend toward axial flow, while the Kaplan turbine includes a vaneless bend downstream of the vanes followed by an axial rotor. All, of course, have axial exit flow. References show that both types have peak efficiencies near 93 percent. In the small sizes, radial turbines have demonstrated an efficiency advantage over axial turbines. Small radial turbines generally look like the section shown in figure 5. Flow enters the vane section from a duct that can be in the form of a torus, which is a doughnut-shaped large plenum, or a volute that provides flow spiraling into the vanes with a significant whirl velocity component. Vanes designed for a volute inlet generally have little or no camber. In most small radial turbines the solid hub does not extend out to the tip as shown. The blades then have running clearance on both the hub side and the shroud side.

Radial or swept blades may be employed. In most applications radial blades are selected because of unacceptable bending stresses in the case of swept blades. A typical velocity diagram is also shown in figure 5 along with that of an axial stage designed for the same speed, work, and work factor. The two significant differences are in the optimum incidence angle indicated by the radial rotor inlet relative velocity vector and the lower exit wheel speed corresponding to the lower exit mean radius. The optimum incidence angle provides minimum inlet loss by positioning the stagnation streamline right on the leading edge. This results in an optimum work factor smaller than 1.0, which limits stage work for applications requiring maximum efficiency. Backsweep and profiled leading edges have been explored for increased stage work and are discussed later in this paper.

An unattractive characteristic of radial turbines is the relatively large volume required by the inlet ducting that lies outside the vane assembly and consequently greatly increases the diameter of the total package. This, of course, also increases the weight. Another is the difficulty of manufacturing a cooled radial rotor for high performance light weight engine systems. Extensive work is underway to develop this technology and is discussed herein.

There are many references describing the general characteristics and the highly efficient performance of radial inflow turbines. One of the most comprehensive and enthusiastic is Homer Wood's paper of 1962, reference 2. This is recommended reading for any designer considering the selection of radial or axial stages for his application. There is a persuasive argument for the selection of higher tip speeds and high stage loadings for the radial in view of the relationships among blade stress, disc stress, efficiency, and stage loading. The radial turbine chapter in NASA SP 290, reference 3, discusses many fundamental considerations in the design of radial turbines with particular emphasis on effects of changes in geometry.

Efficiency

The radial gas turbine is in competition with the axial turbine mostly in the small size range. Here its packaging and weight disadvantages are offset by the inherent advantages in efficiency. Small axial turbines are very sensitive to aerodynamic compromises associated with manufacturing limitations and inaccuracies. These compromises are in the areas of airfoil thickness, surface roughness, rotor tip clearance control, and general dimensional controls. Reference 4 details a study of a six inch tip diameter axial turbine that performed with as-manufactured rotor blades at an efficiency level of 78 percent at its design operating point. Correcting the surface roughness gained one point. Reworking a large part of the profile to reduce trailing edge thickness by 21 percent further increased efficiency by 4 percent. These changes raised the efficiency to 83 percent. Trailing edge thicknesses were still relatively large, however, with a blockage of about 13 percent, and hurt performance. Another significant penalty is in rotor tip clearance. While large aircraft turbines operate with tip clearances near 1 percent of passage height, small turbines typically operate with tip clearances of 2 to 3 percent. This penalizes efficiency in three ways. The tip section unloads, reducing turning, a significant amount of working fluid bypasses the rotor blades, and there are viscous losses incurred by the tip clearance flows. The tip clearance loss reduces turbine work and efficiency by one to three percent for each percent clearance depending on clearance configuration and rotor tip reaction. For a high reaction rotor tip with a smooth shroud wall, the figure is three.

The radial turbine experiences the same absolute clearance controls and levels, but its flow path provides a very minimal clearance penalty. This is perhaps the key factor in the radial turbine's efficiency advantage over the axial. Minimum tip clearance achievable depends on bearing clearances, shaft excursions, differential thermal and centrifugal growths, and manufacturing tolerances. In general, the clearance gap is proportional to the clearance radius. Note in figure 5 that the radial turbine exducer radius is only about two thirds that of the axial while the blade is about three times as high. This provides a radial clearance to passage height ratio about one quarter that of the small axial turbine and well under the 1 percent achievable in even large axial turbines. This clearance-efficiency relationship was studied with a low pressure radial turbine designed for a space power system and reported in reference 5. The key results are shown in figure 6. Axial clearance was varied near the leading edge and the radial clearance over the exducer with smooth variation between. The sensitivity for the axial (inlet) clearance is only one-tenth that of the radial (exit) in the zero to three percent clearance range. This insensitivity to axial clearance at the large inlet radius results from the low relative velocity and consequently low viscous clearance flow loss. There is no real loss in turbine work because the stator vanes provide the inlet angular momentum while the tight exit running clearance ensures near-design extraction of this angular momentum and consequently near design work. The adverse effects of large axial clearances are thus limited to the clearance flow losses and a slight redistribution of the blade loading. Another radial turbine advantage is the unguided acceleration in the vaneless space. This reduces vane loading and losses, particularly in a case of high stage work where the vaneless space could provide acceleration to supersonic velocities. Two other assets are the light blade loading and the lower average kinetic energy levels. The very high rotor solidities result in relatively light blade loading with low blade surface diffusion and

the low average velocities lead to somewhat lower rotor viscous losses. All of these efficiency advantages stem from the radial-axial flow path illustrated in figure 5.

Current Problem Areas

There are several problem areas being worked by various engine companies, laboratories, and universities. Much of the motivation and funding comes from the Department of Defense and the Department of Energy. Research and Development (R&D) efforts are underway for the future use of radial turbines in applications ranging from helicopters to trucks. The importance of low fuel consumption leads to work toward lighter and more efficient turbines to operate in the high temperature high pressure environment of high performance helicopter engines. This identifies cooling and materials as candidate items for R&D effort. Low cost is important in any application, but critical in ground vehicle engines because of the advanced manufacturing and performance technology of competing diesel and spark ignition engines. This consideration directs efforts toward low pressure ratio systems with recuperated cycles and turbines with little or no cooling. The major concerns being addressed for future engines are as follows:

(1) Increased turbine inlet gas temperatures
- o highly effective cooling with multi-pass coolant passages, turbulence promoters, and some film cooling
- o improved manufacturing technology to provide complex coolant passages at reasonable cost
- o metals that can tolerate high temperatures
- o ceramic coatings
- o solid ceramics

(2) Improved part power performance
- o better aerodynamics
- o variable geometry

(3) Improved duct design
- o 2-D axi-symmetric viscous flow codes
- o 3-D viscous flow codes

(4) Increased stage loading
- o blade backsweep
- o downstream stators

(5) Improved off-design performance prediction
- o better loss models
- o variable geometry
- o cooling effects

Recent and Current R&D Programs

A variety of programs in radial turbine technology have been carried out in recent years with varying degrees of success relative to their objectives. Topical areas included air cooling, materials, fabrication techniques, and variable geometry. Many of these efforts were funded by the U. S. Army with

engine requirements derived from helicopter mission studies. These studies have shown that for engines with flows of five pounds per second or less, a radial stage is a very attractive component for the compressor drive. It packages well with a centrifugal compressor and reverse flow annular combustor, and frontal area is not of serious concern. Most applications require front-drive power turbines with concentric shafts. This means fairly large bore diameters in the core compressor and turbine. High bore stresses require high mechanical integrity of the radial turbine rotor. Much of the development work for high temperature applications with air cooling, therefore, have been in this area of mechanical strength and durability in the rotor hub as well as in the blades.

<u>Bicast cooled rotor</u>. - One of the early programs for developing a cooled radial turbine was initiated with funding by U. S. Army AVLABS in 1968 (ref. 6). This program called for development of a design and fabrication method for an air cooled rotor for an engine air flow of 5 pounds per second and a turbine inlet temperature of 1530 K (2300° F). The turbine tip speed was 2300 feet per second and expansion ratio of 5:1. This rotor had a relatively small bore; smaller than current front-drive requirements. The blade cooling was relatively simple with a single two-pass channel in each blade discharging the coolant through a radial slot on the blade surface as shown in figure 7. Initial attempts to cast a one-piece rotor were not successful, with gross nonfill, ceramic core breakage, and distortion. Subsequent attempts involved bicasting, figure 8. This was unsuccessful also, however, with insufficient rotor strength to support the high speed stresses.

<u>Cast/HIP cooled rotor</u>. - Detroit Diesel Allison (DDA) was awarded an army contract in 1977 to provide and demonstrate the technology required for the economical manufacture of an efficient high temperature radial turbine (refs. 7 and 8). The approach selected included an air-cooled airfoil shell HIP bonded to a high strength hub. The shell was a monolithic casting that included all rotor blades and a thin hub section. The bond joint surface was conical under most of the blade section with a short cylindrical surface under the exducer section. The cooling scheme was relatively simple with two smooth-walled channels discharging air on the pressure surface of each blade near the trailing edge. The cooling passages and the HIP (Hot Isostatic Pressing) bond surfaces are shown in figures 8 and 9.

The aerodynamic and thermal design involved a tradeoff study among many variables. These included blade-jet speed ratio for high efficiency, blade and endwall contouring, coolant flows, and various stresses. With an engine output of 1000 horsepower, the core turbine was designed for 55 000 rpm, 1258 hp, 170 psia, 1530 K (2300° F), and 5.2 lb/sec. An excellent bond quality was achieved. This was demonstrated by ultra-sonic non-destructive testing, macro/microstructure examination of a sectioned rotor, and rotor-burst strength testing. Production cost estimates made this approach very attractive relative to the cost of a comparable axial turbine.

Low temperature aerodynamic performance tests were run over a range of speeds and pressure ratios. Varying coolant flow rates from 0 to 150 percent of design flow ratio were run in order to learn the effect on performance. Figure 10 shows a performance map with efficiency referenced to only the primary flow. Design point efficiency of 86 percent is somewhat short of the 88 percent goal. The goal should be achievable in a developed turbine. Figure 11 shows the effect of coolant flow on thermodynamic efficiency defined as the

ratio of output work divided by the sum of primary and coolant flow available energies.

Laminated radial wafer cooled rotor. - References 9 and 10 describe an Army contract effort at the Garrett Turbine Engine Company. This effort paralleled the DDA program with a different fabrication approach. A large number of thin photo-etched sheets of MAR-M 247 were stacked axially and HIP-bonded to form the rotor. The external surfaces forming the blades and the hub were then machined to provide smooth flow passages. Photo-etching permitted the use of complex cooling passages formed by the etched openings in the laminates. These openings were normal to the laminates giving rise to the steps shown in figure 12. The discontinuities enhance internal heat transfer by promoting turbulence, but they result in thick blades in the exducer region because of the discontinuities. This turbine also included a separately formed hub bonded to the laminated assembly. This program experienced many difficulties in the achievement of high integrity bonds between the laminates and also in the bond inspection process. The radial wafers support centrifugal stresses well, however, and the test rotors exhibited good burst strength. The problems experienced indicate a limited future for this design approach.

Split blade concept. - A program jointly funded by NASA and the U. S. Army Research and Technology Laboratories is currently underway at Solar Turbines, Inc. This concept, shown in figure 13, employs two monolithic castings, a "star wheel" inlet section, and an exducer section. The "star wheel" is cast with split blades. An iron core for each blade is formed with grooves and cavities to be filled with the same high temperature alloy used for the blade-wheel casting. This is HIP bonded into the split blades. The iron is subsequently leached out with acid leaving complex internal cooling passages with flow boundaries and turbulence promoters where the iron core had been grooved. This technique permits the use of highly effective internal cooling and also flexibility in the internal geometry. Modifications may be made readily in the iron matrix with no reworking of the casting patterns or cores. The exducer casting can be produced with highly effective cooling passages and with no difficult casting problems. It's too early in the program to assess the success in meeting objectives, but the concept is attractive. Figure 14 shows an exducer at the top and the split blade casting at the left. The casting shown on the right is a company-funded alternative to the split blade approach. It is a cored casting. Both of these "star wheels" discharge the coolant from the trailing edges into a gap between the radial part of the blades and the leading edges of the exducer blades forming a cooling film on the exducer blades. The exducer blades themselves are internally cooled with trailing edge ejection. It should be noted that the design requirements are rather severe with a turbine rotor inlet gas temperature of 1800 K (2800° F). A paper describing this program in some detail is being prepared for presentation by A. Hammer of Solar at the 1983 SAE Aerospace Meeting.

Variable geometry. - The need for improved part power fuel consumption led the U. S. Army Research and Technology Laboratories to award two study contracts in 1979. The purpose was to define approaches in variable-geometry components that would provide an output power range of 2:0 while maintaining a nearly constant thermodynamic cycle. The full power flow was specified at 5 lb/sec and two turbine inlet temperatures were to be examined, 1650 K (2500° F) and the maximum attainable with an uncooled rotor. One of these studies,

performed by Teledyne, resulted in the concept shown in figure 15. Each of two centrifugal compressor stages is equipped with variable inlet guide vanes and moveable diffuser sidewalls. The radial core turbine also employs a moveable stator sidewall while the axial free power turbine has pivoting stator vanes which, like the compressor inlet guide vanes, rotate about the radii of concentric spherical endwalls. The rotor blades are not cooled. All of the variable components are linked mechanically to simultaneously vary the flow area through the range of 50 to 100 percent of the full power flow area. Shaft speeds and turbine inlet temperature are to be held constant so that all component pressure ratios and flow-specific works are also approximately constant. This results in a net power output that also varies from 100 down to 50 percent, except, of course, for whatever performance penalties are incurred by the variable geometry. This study contract effort led to additional contracts for the design, fabrication, and experimental evaluation of a variable radial inflow turbine with various mechanical approaches to the translating endwall. These included straight and contoured endwalls on the hub side and the shroud side. Each of these configurations, 28 in total number, involved stationary vanes and one moveable endwall with holes matching the vane profiles to permit the translating movement. So far in the program, 24 configurations have been tested. For the most efficient, 50 percent area reduction resulted in an efficiency decrease from 87.5 percent to 83.5 percent and a flow reduction of 38 percent for a net power reduction of about 40 percent. Figure 16 shows one of the preliminary data plots for a moveable hub wall. This curve shows a somewhat lower part power efficiency than the best moveable shroud configuration, but the shape of the curve is typical. These data were taken with the vane-profile-wall clearances completely sealed. Some data taken with a workable high temperature seal installed indicate that leakage flows and aerodynamic losses can be held to very low and acceptable levels. It is likely that the compressor stages and the axial power turbine will exhibit similar characteristics. That is, decreasing efficiency and less than 1:1 flow/area reduction as flow areas in the static vanes and diffusers are reduced. Consequently, engine power may be expected to decrease to a level near 50 percent of full power as fuel flow is controlled to maintain core shaft speed as flow areas are reduced by 50 percent while specific fuel consumption increases reflecting the lower component efficiencies. A paper describing the experimental program as well as the system study background is being prepared for the 1983 SAE Aerospace Meeting.

A second contract study was carried out by the Garrett Turbine Engine Company. An early decision to pursue the cooled rotor approach was made, with a laminated radial wafer fabrication assumed. The variable geometry in this turbine was simply a pivoted trailing edge section on each vane. The stationary parts of the vanes included contoured endwalls for low aerodynamic loss. The flow path and the articulated vanes are shown in figure 17. The large vane exit flow angle results in a required angular movement of only 10° for the 50 percent area reduction. Flat parallel endwalls should facilitate the action of the moving trailing edges with minimal leakage problems. Although this program was not carried through aerodynamic testing, a detailed analysis with good loss models indicated a stage efficiency of 0.866 including stator leakage and all cooling penalties. Reference 11 is the final report on the program.

Both of these approaches are promising and worthy of continued attention.

Ceramic turbines. Ceramics for various turbine engine components are receiving a great deal of attention because of the potential multiple benefits

in cost, weight, corrosion and erosion resistance, and the elimination of cooling in high temperature areas. Reference 12, published in 1970, discusses many of the design considerations in the use of ceramics, makes comparisons with high temperature metals, and includes many references on the subject that were published during the 50's. Some of the problem areas detailed therein have been worked extensively with some measure of success. Reference 13 provides a more recent overview of the subject with a summary of R D program efforts and progress. This paper assessed fairly well the state of the art and the problems that must be solved to enable exploitation of the potentials of ceramic components. Some of the conclusions reached were as follows:

(1) Large utility gas turbines offer a great potential for conservation of current fuels as well as the potential for using lower grade, more corrosive alternate fuels.

(2) Use of solid ceramic components in aircraft is still pretty far into the future because of high risk. One exception might be the static shroud over the high pressure turbine rotor blade tips.

(3) Earliest use of solid ceramic turbines will be in small sizes for application in automotive passenger vehicles, trucks, buses, remotely piloted aircraft, and portable power units.

(4) R D needs are in improved materials and processing, design criteria and approaches, and non-destructive evaluation techniques.

The Department of Energy has been and is currently funding programs in the development of components for ground transport vehicles. Two of these are the CATE (Ceramic Applications in Turbine Engines) and AGT (Advanced Gas Turbine) projects. Work to date on monolithic ceramics has shown that these materials have good high temperature strength and good oxidation resistance, but they are brittle and currently have low reliability. Maximum operating temperatures for various parts made of silicon carbide and silicon nitride range from 1600 to 1900 K (2400° to 3000° F) depending upon stress levels. Processing begins with powders of submicron particles which are sintered with various additives. Hot isostatic pressing, injection molding, or slip-casting is then employed to form the "green" bodies. Densification by sintering and, again, hot isostatic pressing finally forms the fully dense monolithic ceramic body. The densification process results in an overall linear shrinkage of about 17 percent. Ceramic axial and radial turbines are being developed for the AGT project by the Garrett Turbine Engine Company working with Ford (AGT 101) and Detroit Diesel Allison working with Pontiac (AGT 100). The actual processing of the ceramics is being carried out by companies including Ford, AiResearch Casting Company, Carborundum, and GTE Laboratories.

Figure 18 shows the ceramic parts under development for the AGT 100. The radial turbine rotor was fabricated by Carborundum of silicon carbide. An alternate rotor of silicon nitride is being developed by GTE Laboratories. Reference 14 provides 1982 status of the AGT 100 program and reference 15 covers the AGT 101 effort in ceramics.

Improved off-design performance. It was mentioned earlier that most radial turbine rotors have thin radial blades at the inlet. This feature leads to a fairly limited range of low-loss incidence angle and consequently high losses at off-design operating points. The Cummins Engine Company has

carried out an interesting program with a series of rotors with thick leading edges shaped to expand the range of low-loss incidence angles. The application associated with the need for this program is in supercharger turbines for large diesel engines. These turbines must operate in pulsating engine exhaust flow over a wide range of engine operating conditions. At any particular engine operating point the mass flow pulsates as much as 20 to 30 percent. During these pulsations a thin blade rotor, upper left, figure 19, develops large intermittent separated recirculation zones. Several rotors with thick blade leading edges were designed, built, and tested for aerodynamic performance. Three of these, round nose, forward curved, and backward curved, are also shown in figure 19. Aerodynamic performance of all three is described in reference 16. Tests used to determine this performance were run at a number of steady flow conditions representing various instantaneous operating points in the real engine environment. Significant improvements in flow capacity and efficiency were observed. Maximum efficiency improvement was [illegible] percent at the high work factor end of the range and 18 percent at the low end. These improvements were for, in Cummins terminology, forward curved and backward curved rotors respectively. The round nose rotor was intermediate between the thin blades and the preferentially curved blades and experienced most of the performance gains at both ends of the range. Subsequent tests in an engine exhaust system verified the performance gains measured in the steady flow tests. The thick blade concept can be adapted to and optimized for a variety of applications for which the conventional geometry is not optimum. Stresses would be more manageable than those of a thin swept blade and design velocity diagrams could be varied freely.

Production Turbines

Most radial turbines that are currently in production are small and used principally in superchargers, aircraft APU's, and aircraft starters. There are some exceptions. Turbonetics Energy, Inc., a subsidiary of Mechanical Technology, Inc., has radial steam turbines in production with output power ratings of 500 to 5000 horsepower. These are suitable for cogeneration, enhanced oil recovery cogeneration and mechanical drive applications. Rotor tip diameters up to 12 inches have been designed with operating speeds of 24 000 to 46 000 rpm. Stage efficiencies of 88 percent have been achieved. Figure 20 shows one of their rotors. Their system includes two radial turbine stages in series coupled to a common output shaft through step down gears.

Another noteworthy application of radial turbines is offered by Kongsberg of Norway. Their KG 2 engine has a centrifugal compressor and a radial inflow turbine mounted back to back and cantilevered on a single shaft. This shaft drives the generator through a two stage gear reducer. Output power ranges from 1000 to 1700 KW with optional gaseous or liquid fuel systems. The turbine tip diameter is about 24 inches. Figure 21 shows a section through the engine and also a photograph of the rotor assembly.

Solar Turbines, Inc. has produced a large number of Spartan gas turbine engines employing radial turbines. These were relatively small, in the 350 to 400 horsepower range. While no longer in production, these engines are in use world wide. There are very large radial turbines running on blast furnace gas in Russia. Output powers range from 8 to 13 MW. Radial turbines were selected for their ruggedness and erosion resistance in this very dirty environment.

Mixed Flow Turbines

Mixed flow turbines are mentioned briefly here because there is some current effort related to possible aerospace and automotive applications. It appears that a mixed flow turbine can offer the efficiency advantages of a radial turbine. Also, blade backsweep for higher stage work can be incorporated with radial blade elements and consequently no centrifugal bending stresses. This provides the best features of both the axial turbine and the radial turbine. A program at the NASA Lewis Research Center includes a ceramic mixed flow turbine as a candidate compressor-drive for an advanced high temperature rotorcraft engine. Other candidates are a small cooled high work axial stage and a cooled metallic radial stage. The mixed flow stage is still in preliminary design, but will look like the configuation shown in figure 22. The use of backsweep with thin blades is essentially unlimited by centrifugal bending stresses, so that stage loading can be increased substantially beyond that of a comparable radial stage. Tip speeds of 2000 feet per second are reasonable for this geometry with either silicon nitride or silicon carbide. Mixed flow stages are suitable and are being considered for a single-shaft automobile turbine engine as well as small powerplants for target drones and other remotely pilotted aircraft.

AXIAL TURBINES

Major Concerns and Problem Areas

A continuing dialogue with each of the major aircraft engine companies is carried out by various segments of the Lewis Research Center staff. In the areas of turbine design and technology this dialogue has identified a number of uncertainties in the various elements of turbine design and the prediction of aerodynamic and thermodynamic processes. These are listed in figure 23.

Basic boundary layer behavior. - The need to minimize engine weight and the cooling flows that penalize cycle thermodynamic efficiency has led to highly loaded vanes and blades with low solidities. The need for low specific fuel consumption has resulted in high cycle pressure ratios and high turbine inlet temperatures, particularly in high bypass engines for passenger and cargo aircraft. The high core pressures and relatively low flows resulting from the high bypass ratios lead to high turning vanes and blades as well as low passage heights and consequently strong boundary layer flows. The high loading, high turning, and small passage heights compound the difficulties in predicting boundary layer flows. The unsteadiness associated with combustor flows, wakes, and rotation affects the transition from laminar to turbulent flows, making it unsteady with development of an unsteady transition region. Some experiments have shown heat transfer rates intermediate between laminar and turbulent predictions. Separation bubbles with reattachment are not unusual in highly loaded airfoils. Much work must be done to develop good prediction techniques for these conditions as well as the influences of turbulence, pressure gradients, surface curvature, and the addition of film cooling flows.

Secondary flows. High turning and low aspect ratios in turbine passages enhance the development and strength of cross-channel boundary layer flows. The horseshoe vortex formed on each endwall just upstream of the leading edge provides a major redistribution of inlet wall boundary layers, with both legs

of the vortex moving to the suction-surface-endwall corner. Some experiments in a 2-D cascade of high turning vanes showed with an ink-dot technique that all of the inlet boundary layer flow close to the wall is wrapped up in this vortex. All of the endwall flow downstream of the vortex is fresh, developing from the free stream flow and flowing from the blade surfaces. This helps explain some past experience with unpredicted endwall hot spots. Rigorous 3-D viscous computer codes will be required to make these phenomena predictable.

Trailing edge losses and deviation. - The high turning angles in the core and fan drive turbines of current engines are presenting problems because of high blockage and high flow area sensitivity to small changes in blade angle. Better prediction techniques are needed to reduce current development time and cost.

Reynolds number effects. - Low pressure turbine stages with highly loaded vanes and blades are sensitive to Reynolds number effects. Some engine manufacturers have reported rather severe performance losses in the latter stages at high altitude cruise conditions. Reynolds numbers based on axial chord drop to values in the 50 000 range where laminar separation and the prediction of transition begin to be problems. Some cascade and full stage research with these critical airfoils is needed to provide a better understanding of airfoil behavior at these very low Reynolds numbers.

Tip clearance losses. - Tip clearance losses increase rather rapidly with increasing clearance gap as mentioned earlier. The losses are of particular concern to designers of small turbines because running clearances are relatively large. While there is no shortage of loss-gap correlations for conventional shrouded and unshrouded blades, there is a shortage of good ideas for minimizing losses. Some work is underway in various locations to examine and evaluate unconventional geometries in the tip region. These include active clearance control devices, non-uniform work distribution, winglets, clearance recessed in the static shroud, and tip treatments that include wall grooves. These approaches have shown promise, and it is likely that different combinations can be optimized for design cases with various gaps, blade row reaction, and blade loading

Accuracy of heat transfer predictions. - Uncertainties in the knowledge of the external blade environment and boundary layer behavior necessarily result in uncertainties in the prediction of external heat transfer. Similarly, coolant side heat transfer prediction is difficult because of uncertainties in actual flow rates and coolant conditions as well as the behavior of the coolant in a rotational field, and as it passes through the entrance to the blade cavity, through the sharp turns, and past the turbulence promoters. There is a great need for improved research instrumentation, better experimental correlations, and, of course, 3-D viscous flow codes.

Disk cavity flow and heat transfer. - Current technology does not provide precise knowledge of the cooling flows, leakage flows, circulatory flow patterns, and heat transfer in the disk cavity. Consequently, experience factors play a large part in the design and development of turbine disks. Most designers agree that disks are currently overdesigned and therefore heavier than necessary. Also, cooling flows are kept on the high side to avoid ingestion of hot gas into the cavity at the blade roots. This penalizes the cycle thermodynamics and also degrades vane and blade root aerodynamics. Another major consideration is the control of blade tip clearance as the disk, blades,

and shroud change dimensionally because of thermal expansion and centrifugal forces. Research is needed to permit reductions in leakage flows and disk weight to minimum allowables.

Current Engine Technology

NASA has funded several engine design studies in recent years in two major programs for civil aircraft. One of these programs, the Energy Efficient Engine (EEE or E^3) was for large commercial passenger aircraft, while the other was the Quiet Clean General Aviation Turbofan (QCGAT) for small aircraft. Both programs were chartered to define the engine cycles, components, and technology needs for future aircraft with good fuel economy, low noise, and low pollutant emissions.

Energy efficient engine. - General Electric and Pratt & Whitney were awarded contracts in this program and have progressed through component demonstrations. These programs have been well documented and publicized. Figure 24 shows a sectional view of the General Electric concept. The flow path, cycle numbers, and key features are shown. The first core turbine stage has a blade height of only 1.6 inches, demonstrating the effect of the high cycle pressure ratios in even a very large engine such as this with a takeoff thrust of 36 000 pounds and an overall nacelle diameter of about 7.9 feet. The second stage has a blade height of 2.7 inches and a tip diameter of 30 inches. The turbine inlet temperature and high rotative speed combine to require a total core turbine cooling of 18 percent of core compressor flow. High turning angles were employed with a first stator exit flow angle of 74° from the axial direction. Second stator exit flow angle was 69°. The first stage vane cooling included impingement and film cooling of the leading edge and pressure surface. There is also trailing edge ejection. The first stage rotor blades included two multipass cooling passages with limited amounts of leading edge film cooling and with tip and trailing edge ejection. Turbulators were used in the cooling passages. The second stage was cooled without surface film cooling.

The GE low pressure low speed turbine has five stages to drive the fan and the 1/4 booster stage. Airfoil internal cooling is used in only the first vane. Vane exit angles are 61° to 64° in the first four stages and 56° in the last. Rotor turning is around 110° in the first four and 74° in the last. Aspect ratios are high in all vane and blade rows.

The Pratt & Whitney engine is significantly different. A low pressure compressor on the fan shaft is used, the core turbine has only one stage, and the two shafts rotate in opposite directions. This engine also has a sea-level thrust of 36 000 pounds, and the nacelle outer diameter is about 8.7 feet. Core turbine cooling is 14 percent of core flow. Vane turning is very high, 78°, and the stage work factor, defined as the change in whirl velocity divided by wheel speed, is 1.6. Rotor turning is 120° and an exit whirl of -44° sets up prewhirl for the counter-rotating low pressure turbine. Vane cooling includes three passages with impingement and film cooling of the leading edge and the pressure surface. The trailing edge has internal turbulators and trailing edge ejection. The rotor blades employ multipass passages with internal turbulators. Coolant leaves the blades along the leading edge for film cooling, the tip pressure surface, and trailing edge. Reference 18 has the details on this turbine.

The low pressure turbine has only four stages, with 15° of turning in the first vane. All rotors have about 110° of turning. An exit guide vane was designed to accept a 40° swirl angle from the last turbine stage and turn the flow to the axial direction with a deceleration of about 20 percent.

The overall engine design and the turbine design in particular have demonstrated a significant step in the achievement of another general goal of engine designers. That goal is the reduction of cost and weight through the reduction in numbers of stages and airfoils.

Quiet clean general aviation turbofan. Two companies, AVCO Lycoming and Garrett, were awarded contracts in 1976 in this program of cycle study, engine design, and engine demonstration. The program goals were to demonstrate significant reductions in noise, pollutant emissions, and fuel consumption from the levels characteristic of airplanes flying at that time. AVCO Lycoming undertook a very ambitious effort that departed significantly from the technology of then current engines. For a flight Mach number of 0.6 at an altitude of 25 000 feet AVCO selected the cycle shown in figure 26. Note that the bypass ratio is higher than those of the E³ engines while the fan pressure ratio is lower. This engine was very quiet with low emissions, bettering program goals in those areas. While fuel consumption exceeded the goal, the reasons were in the area of component development. The program did not have allowance for this. This is a fairly small engine with a sea level static thrust of 1600 pounds. The nacelle outer diameter is about 21 inches. Both the core turbine and the power turbine are one stage machines with a gear box speed reducer between the power turbine and the fan. The core turbine stage work factor was on the high side at 1.8, and cooling was required for both vanes and blades. The rotor blade height was 0.66 inches at the leading edge, 0.86 inches at the trailing edge. Vane cooling was accomplished with two pass convection cooling and ejection on the pressure side of the trailing edge. Pin fins were used as turbulators in the nose and trailing edge sections. No film cooling was used. The rotor blades of the core turbine were cooled in about the same way but with a split trailing edge.

The power or low pressure turbine is uncooled. The shaft speed, 32 000 rpm, is well below that of the core turbine, 49 000 rpm, but high enough to support the single stage design. The work factor is 1.5. Reference 19 is the final design report.

The Garrett engine is shown in figure 27. It was designed for a cruise Mach number of 0.8 at an altitude of 40 000 feet. The bypass ratio, 3.7, is considerably lower than AVCO's with a geared fan pressure ratio of 1.6. The engine is a modification of an existing turbofan engine. This was therefore a lower risk enterprise. Bypass ratio was increased slightly from that of the base engine. Other changes included adaptation of a fan from a different engine, a new gear box, a new low pressure turbine, and a mixer compound exhaust nozzle. This engine met or exceeded all of the major program goals. This engine is somewhat larger than the AVCO engine with a sea level static thrust of about 4000 pounds. The nacelle diameter is about 38 inches. No detail on the core turbine was included in reference 20 because of proprietary considerations and the fact that it was an existing design and not a product of this program. The three stage low pressure turbine is fairly conventional with shrouded rotors and stage work factors of 1.1 to 1.5.

Long Range Needs in Turbine Technology

Improved aerodynamic and thermodynamic turbine design tools are needed to decrease the life cycle costs of operating engines. These tools can reduce development costs, increase hot part life, and reduce fuel consumption. There has been an awareness of these specific needs for many years as incremental improvements in design techniques were continually developed. Future improvements will depend upon improved instrumentation, new and improved computer codes, and the continuing improvement in computers. Figure 28 lists the major needs that have been identified by designers of turbines for large and small engines. 2-D codes for annular ducts and turbomachinery blade to blade flow surfaces are already in wide spread use, and at least one 3-D code has been used with some success by one of the engine companies. Many people and organizations are working on diverse approaches to the 3-D problem. These codes, with various levels of sophistication, will gradually be integrated into the design system. Figure 28 outlines the features that will be incorporated into the codes. In addition to those listed, unsteady flow effects must be handled in some way. The other needs have already been discussed to some extent, and certainly must be included in this list. The last item is the commonly used term to identify blade root centrifugal stress level. This product of annular flow area and the square of rotational speed is directly proportional to blade centrifugal stress given a material density and a taper factor. Increases in currently limiting AN^2 values are desired by all aircraft turbine designers for the improvement of aerodynamic performance. Improved materials and cooling are the keys to improvement here.

Mission and Cycle Studies for Future Engines

Mission analyses and cycle studies are routinely carried out by NASA as well as the aircraft and aircraft engine companies for a variety of civil and military applications. Turbofans and advanced prop fans for flight in the year 2000 are the subjects of some current studies. Advanced technology projections, cycle trends, engine configurations, and operating costs are being analyzed through parametric calculations with ranges of cycle pressure ratio, turbine inlet temperatures, and component efficiencies. Calculations of this nature have been and are being carried out for several flight altitudes, Mach numbers, ranges, and mission profiles.

Figure 29 shows the elements considered for large (60 000 pounds thrust) and small (25 000 pounds thrust) turbofan engines for civil transport. A typical display of cycle study results is shown in figure 30, with thermal efficiency as a function of combustor exit temperature for several component efficiency levels and overall pressure ratios. Overall efficiency is the product of this thermal efficiency and propulsive efficiency, and specific fuel consumption is inversely proportional to the overall efficiency. Each of these efficiency terms is useful in evaluating the total propulsion system. The components represented by these efficiency levels are the fan inner duct, the compressor, and the core turbine. The curves have optimum bypass ratios and fan pressure ratios built in for this flight condition. Note also that the turbine cooling flow was assumed to be zero. The effect of cooling flows can be accounted for by equating the penalty to a loss in turbine efficiency.

The computer code used to generate this fairly simple display is complex and at any point on this matrix partial derivatives or influence coefficients

can be calculated for each variable in the program. Comparisons among these influence coefficients identify the high and low payoff areas, and thus aid in the planning of R D efforts. Curves of this type, along with estimates of future technology, are also used to project future cycle trends. Figures 31 and 32 show a preliminary view of the trends indicated by this study.

It appears that cycle pressure ratios will continue increasing to levels near 60 compared to todays maximum values near 30. Turbine inlet temperature increases, however, will be minimal, with changes of only 55 to 110 K (100° to 200° F). The effect of these changes on the turbine will include smaller passage heights and aspect ratios while higher heat fluxes will result from the higher pressures. Also, the coolant will be hotter. Therefore, both the aerodynamic and the heat transfer design problems will become more difficult. Improved cooling and materials will be needed to avoid cycle performance penalties associated with high turbine cooling flows. All during the history of the aircraft gas turbine increases in inlet temperatures have been paced by materials. As allowable blade surface temperatures increased inlet gas temperatures increased, paving the way for the enormous gains in engine thrust-to-weight ratio and the reductions in specific fuel consumption. Figure 33 shows the evolution of turbine materials as they were developed to provide the required strength at ever increasing surface and bulk metal temperatures. There has been a continuing increase in allowable surface temperatures with occasional discontinuities reflecting "break throughs" associated with new materials and new processes. Two of these, mentioned previously, are projected for ceramic coatings and solid ceramics.

Turbine Performance Goals

Studies made recently and currently in progress have identified performance goals for the next generation of civil aircraft turbofans. Similar studies made for propeller driven aircraft were made. Flight Mach numbers and altitudes were somewhat lower than those specified for the turbofan studies, 0.75 and 35 000 feet for the turboprop. The turboprop studies so far have assumed that the high pressure spool would use essentially the same core compressor and turbine technology as the turbofan. Projections for small engines have also been made for various general aviation and rotorcraft applications. A very brief summary of the turbine performance goals and operating characteristics is shown in figure 34.

Military engines and related technology have not been discussed herein because much of the mission related numbers and the engine operating conditions are classified. In general, thrust-to-weight ratio outweighs considerations of noise, fuel economy, and long life. Turbine temperatures are therefore somewhat higher than in civil engines and transients are certainly more severe. Core turbine technology is not greatly different from that of civil engines, however, and as time goes by there is progressively greater commonality. In the early days of jet engines, military engines led the way with technology transfer to civil engines several years later. That technology transfer has become a two way street in recent years. Much of the work on design methods and turbine materials is led by the designers of civil aircraft engines and in many cases the pioneering efforts are jointly supported. The long range goals in turbine technology for military and civil applications are nearly the same. Achievement of these goals will require continuing efforts in the analysis and design of turbine main flow passages and coolant

channels. Figure 35 shows schematically the projected change in turbine technology from today's approximate and highly uncertain methods of flow and metal temperature prediction to future techniques with greatly improved accuracy in the aerodynamics and heat transfer. The critical element, of course, is the development of rigorous and verified computer codes.

CONCLUDING REMARKS

The material presented here has been gathered from a variety of sources describing many research and development programs. These efforts over the past thirty-five years have resulted in the current high levels of engine and component performance. American dominance of the international aircraft market is now being seriously challenged by several countries and consortia of companies. The improvements in turbine design technology required for continued U. S. leadership in the aircraft engine industry will come largely from a thorough understanding of fluid behavior in turbine passages and the concomitant ability to predict this behavior. Critical elements in the development of rigorous computer codes for hot gas flow field prediction include precise non-interfering instrumentation, thorough experiments for accurate modeling, and experiments in near-engine environments to validate the computer codes. Our work is cut out for us.

REFERENCES

1. Stepka, F. S., "Uncertainties in Predicting Turbine Blade Metal Temperatures," ASME Paper 80-HT-25, 1980.
2. Wood, H. J., "Current Technology of Radial-Inflow Turbines for Compressible Fluids," ASME Paper 62-GTP-9, 1962.
3. Rohlik, H. E., "Raidal-Inflow Turbines," Turbine Design and Application, vol. 3, A. J. Glassman, ed., NASA SP-290, 1975, pp. 31-58.
4. Roelke, R. J., and Haas, J. E., "The Effect of Rotor Blade Thickness and Surface Finish on the Performance of a Small Axial Flow Turbine," ASME Paper 82-GT-222, Apr. 1982.
5. Futral, S. M., and Holeski, D. E., "Experimental Results of Varying the Blade-Shroud Clearance in a 6.02-Inch Radial Inflow Turbine," NASA TN D-5513, Jan. 1970.
6. Calvert, G. S., Beck, S. C., and Okapuu, Ulo, "Design and Experimental Evaluation of a High Temperature Radial Turbine." PWA-FR-4058, USAAMESL TR-71-2C, Pratt and Whitney Aircraft, West Palm Beach, Fla., May 1971.
7. Ewing, B. A.; Monson, D. S.; and Lane, J. M., "High Temperature Radial Turbine Demonstration," AIAA Paper 80-0301, Jan. 1980.
8. Ewing, B. A.; and Monson, D. S., "High-Temperature Radial Turbine Demonstration," DDA EDR-9990, USAAVRADCOM-TR-80-D-6, Detroit Diesel Allison, Indianapolis, In., Apr. 1980.
9. Vershure, R. W., et al., "A Cooled Laminated Radial Turbine Technology Demonstration." AIAA Paper 80-0300, Jan. 1980.
10. Lane, J. M., "Cooled Radial Inflow Turbines for Advanced Gas Turbine Engines," ASME Paper 81 GT-213, Mar. 1981.
11. Large, G. D., and Meyer, L. J., "Cooled Variable Area Radial Turbine Technology Program," NASA CR 165408, Jan. 1982.
12. McLean, A. F., "The Application of Ceramics to the Small Gas Turbine," ASME Paper 70 GT 105, May 1970.

13. Probst, H. B., "Substitution of Ceramics for High Temperature Alloys," NASA TM-78931, 1978.
14. Helms, H. E.; and Johnson, R. A., " Advanced Gas Turbine Technology Development: AGT 100 Systems and Components," 20th Automotive Technology Development Contractor's Coordination Meeting, P-120, SAE, Pennsylvania, 1983, pp. 155-166.
15. Boyd, G. L., et al, "Advanced Gas Turbine Ceramic Component Development," 20th Automotive Technology Development Contractor's Coordination Meeting, P-120, SAE, Pennsylvania, 1983, pp. 189-198.
16. Mulloy, J. M.; and Weber, H. G., "A Radial Inflow Turbine Impeller for Improved Off-Design Performance," ASME Paper 82-GT-101, 1982.
17. Halila, E. E.; Lenahan, D. T.; and Thomas, T. T., "Energy Efficient Engine, High Pressure Turbine Test Hardware Detailed Design Report," R81AEG284, General Electric Co., Cincinnati, Oh, 1982. (NASA CR-167955).
18. Thulin, R. D.; Howe, D. C.; and Singer, I. D., "Energy Efficient Engine: High Pressure Turbine Detailed Design Report," Pratt Whitney Aircraft Group, East Hartford, Conn., 1982. (NASA CR-165608).
19. Schrader, W.; and German, J., "AVCO Lycoming QCGAT Final Design Report," LYC 80-45, Avco Lycoming Div., Stratford, Conn., Feb. 1980.
20. Norgren, W. M., et al, "QCGAT, Quiet, Clean, General Aviation Turbofan Final Design Report," Garrett Report 21-2474(2), Garrett, Phoenix, Ariz., Dec. 1978.

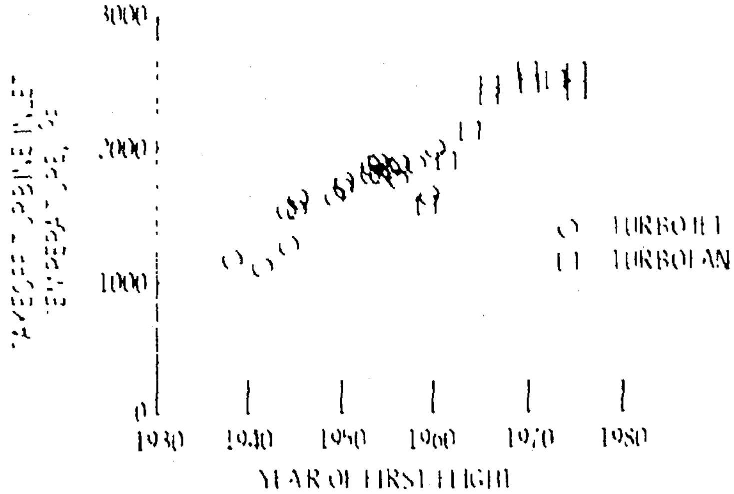

Figure 1. Progress in turbine temperature.

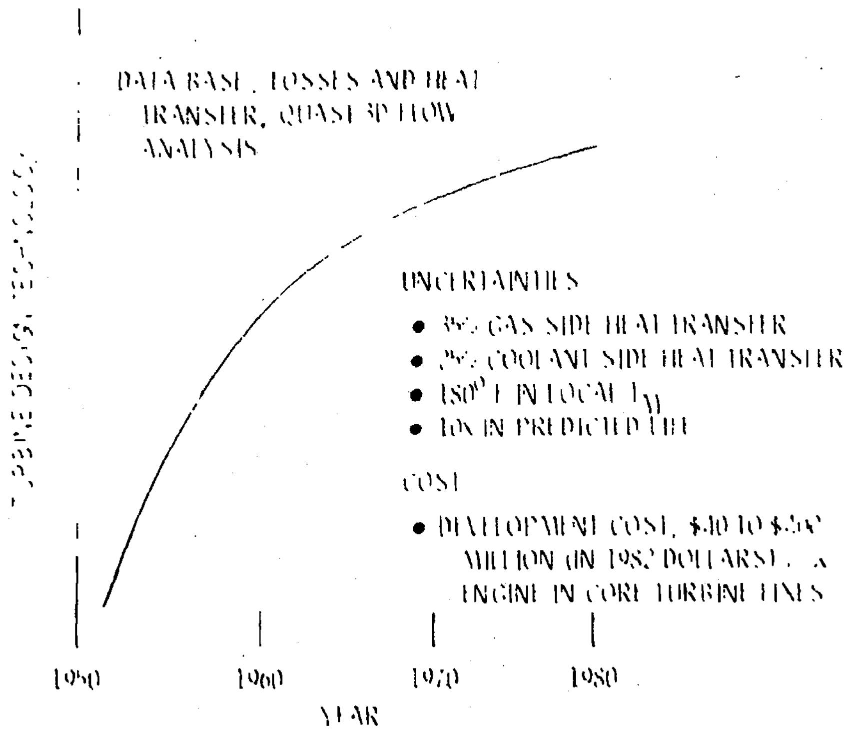

Figure 2. High pressure turbine technology.

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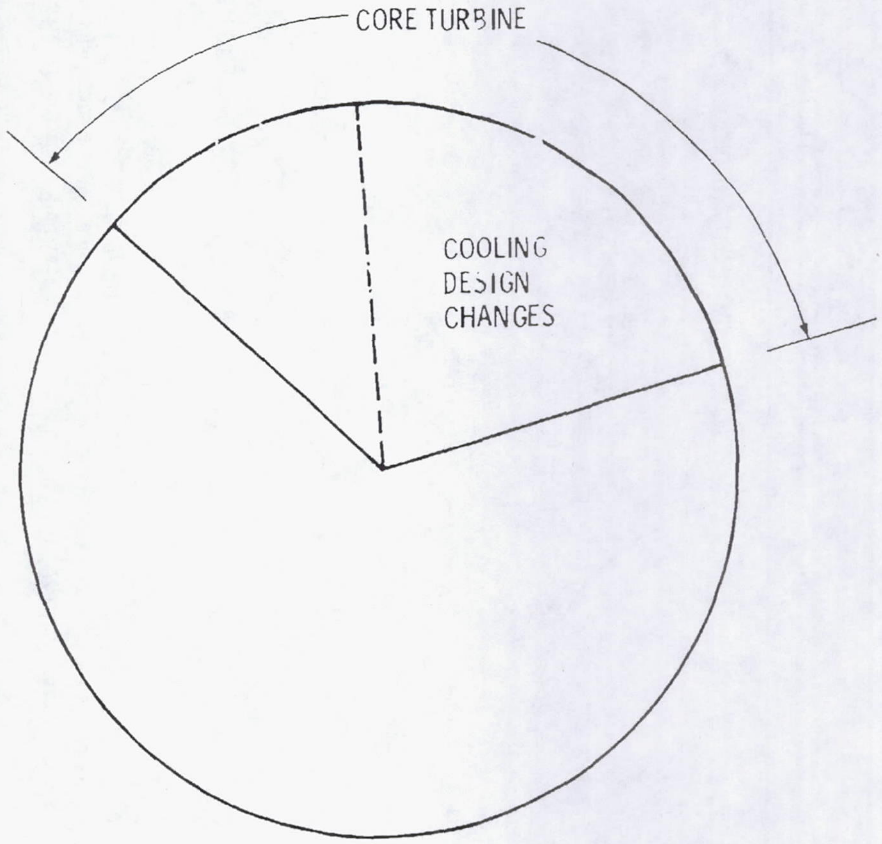

• 600 m TO 1600 m
• 30 TO 40% IN CORE TURBINE
• 2/3 OF THIS IN "FIXES"

Figure 3. - New engine development cost.

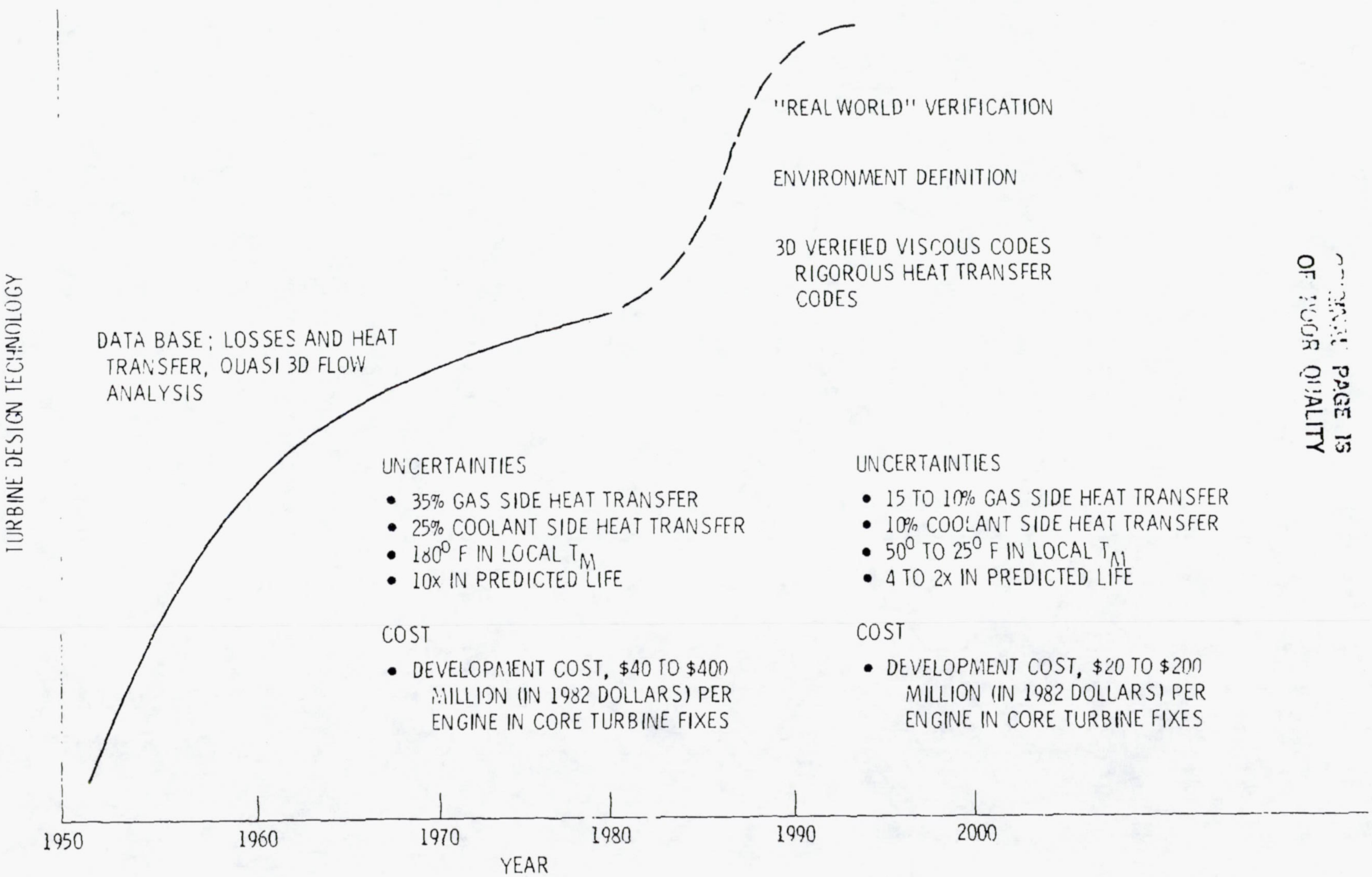

Figure 4. - Past, present, and future technology.

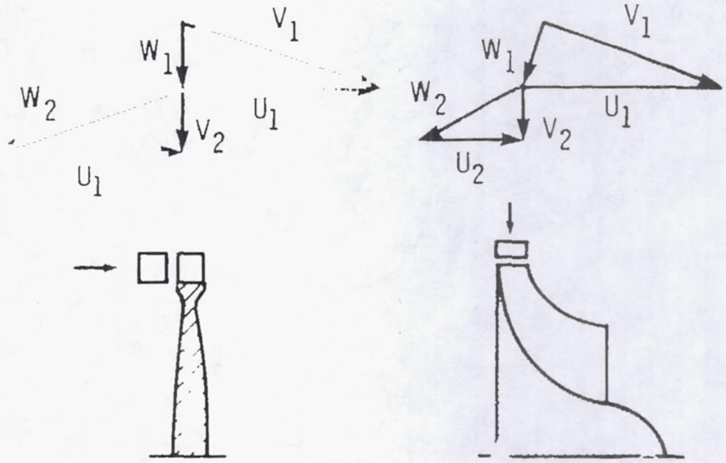

Figure 5. - Axial and radial stages for same application.

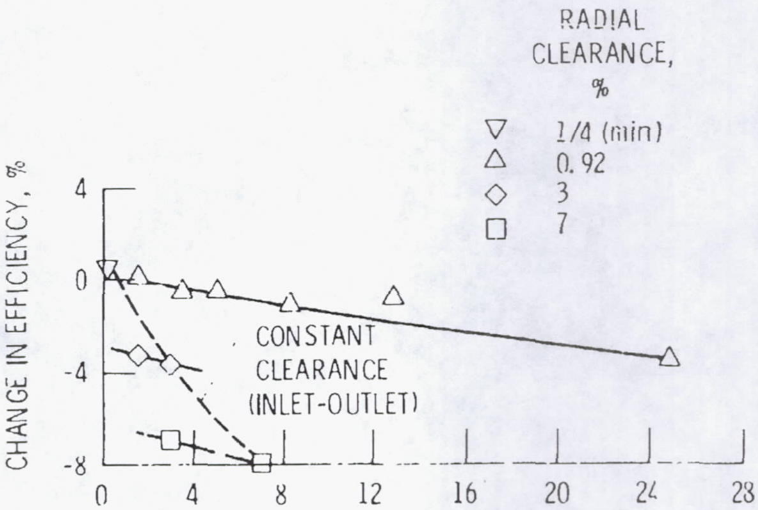

Figure 6. - Radial turbine clearance effects.

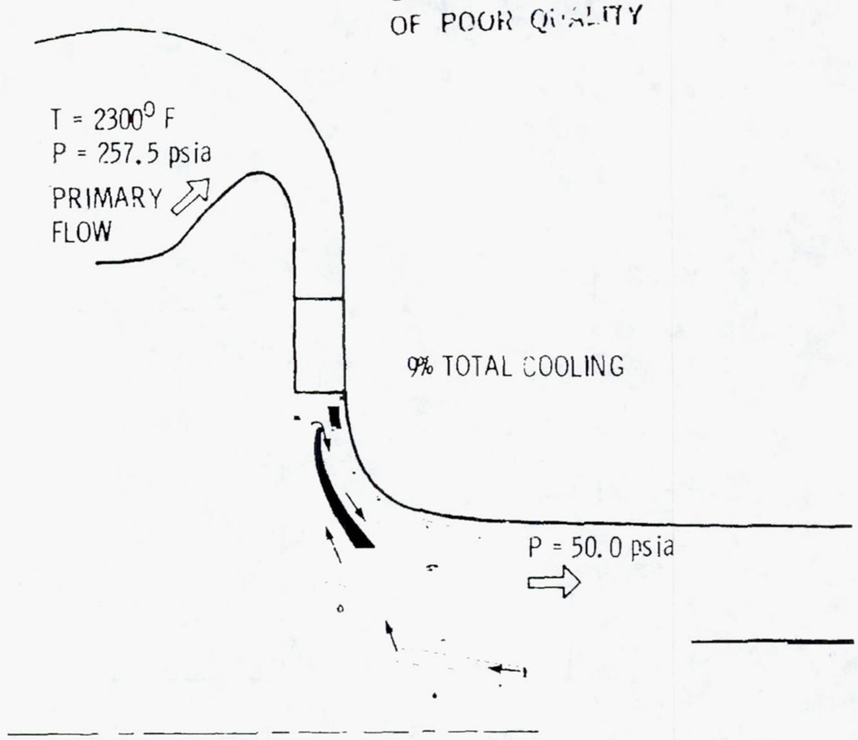

Figure 7. - Cooled radial turbine.

Figure 8. - Bicast rotor section.

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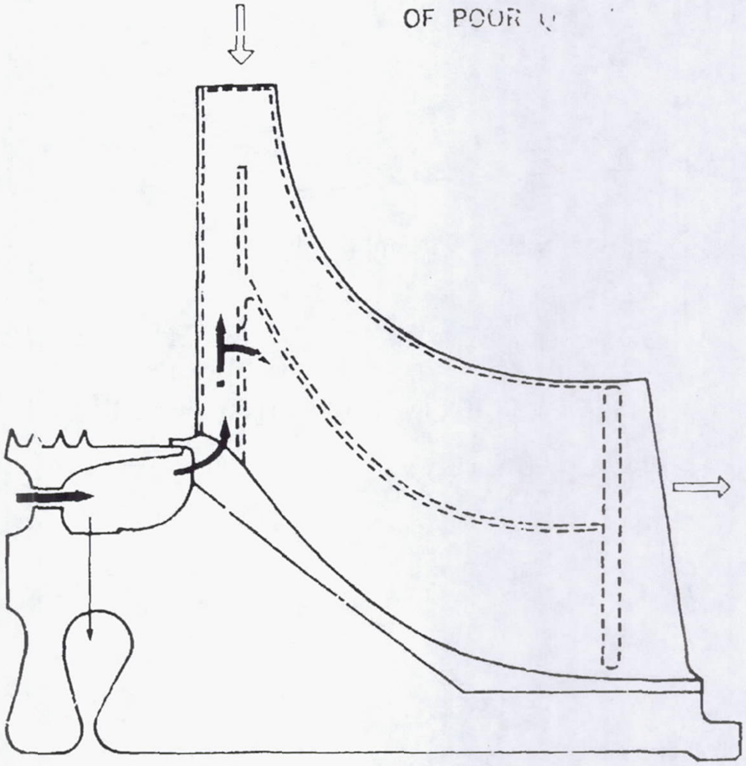

Figure 9. - Cooled radial turbine.

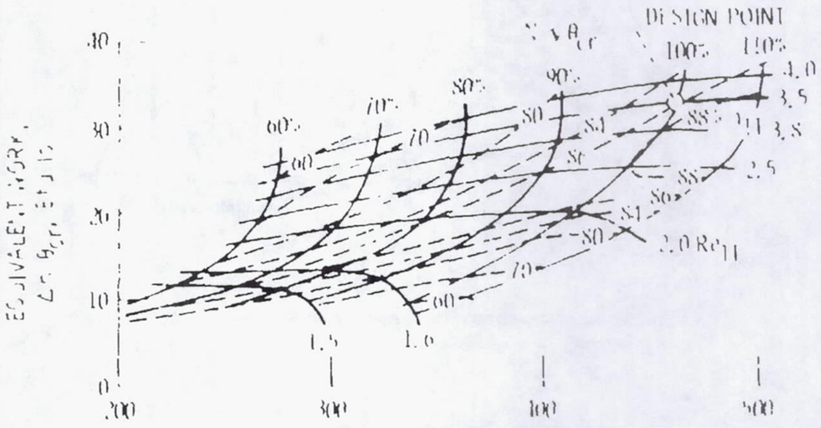

Figure 10. - Turbine performance map.

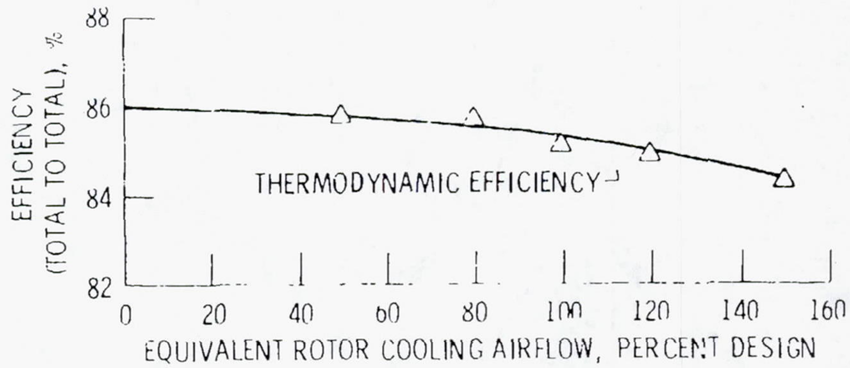

Figure 11. - Efficiency versus coolant flow at design expansion ratio. 100% $N/\sqrt{\theta_{cr}}$.

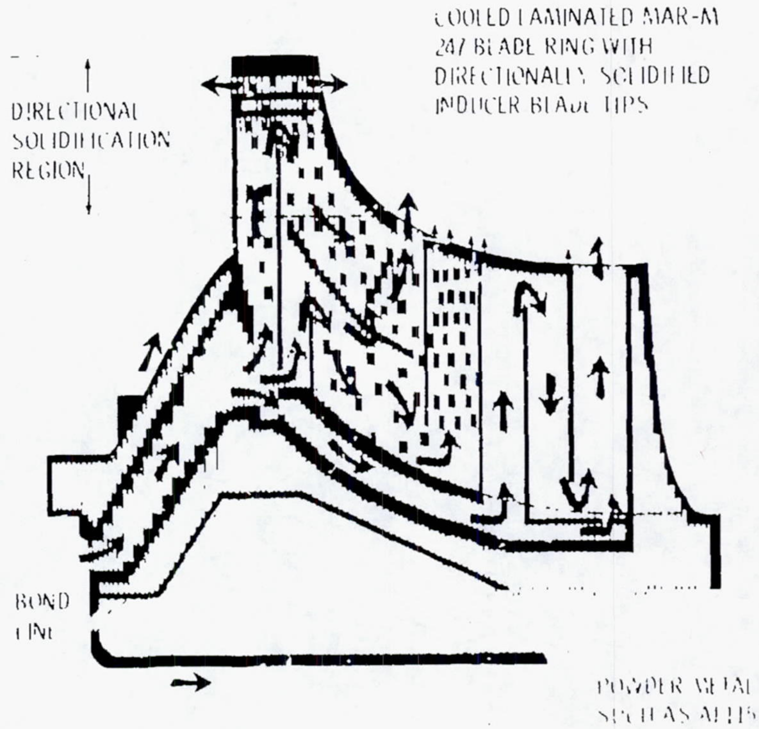

Figure 12. - Cooled laminated rotor concept.

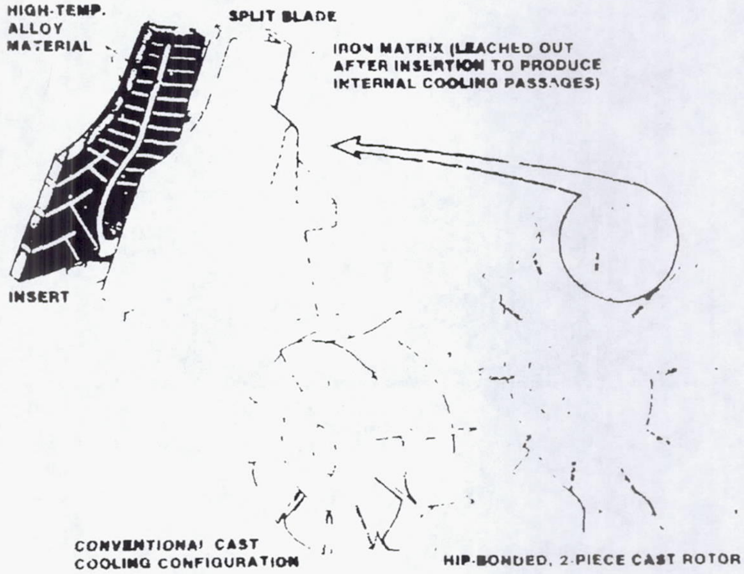

Figure 13. - Fabrication of cooled radial turbine rotor.

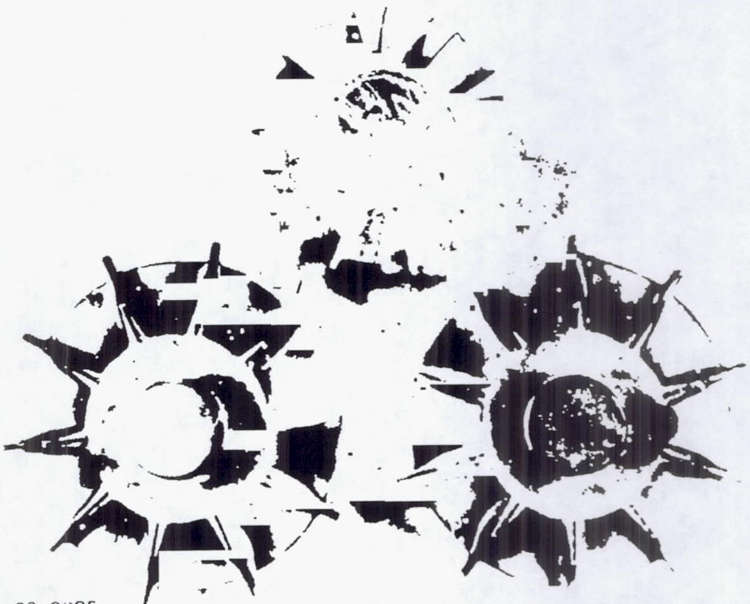

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Figure 14. - Castings of cooled radial turbine rotor.

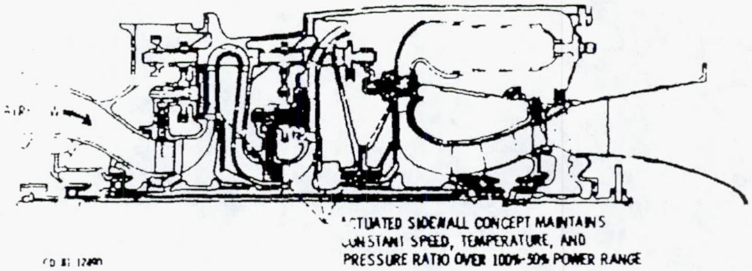


Figure 15. - Army/NASA variable engine-Teledyne.

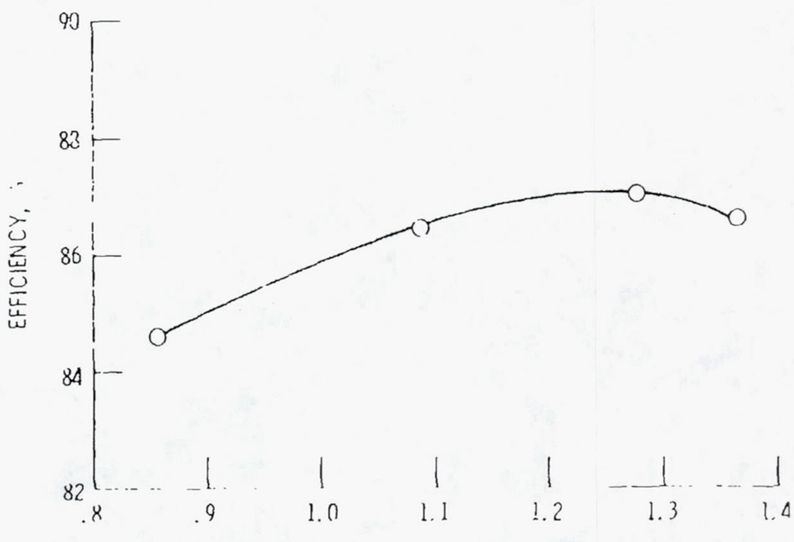
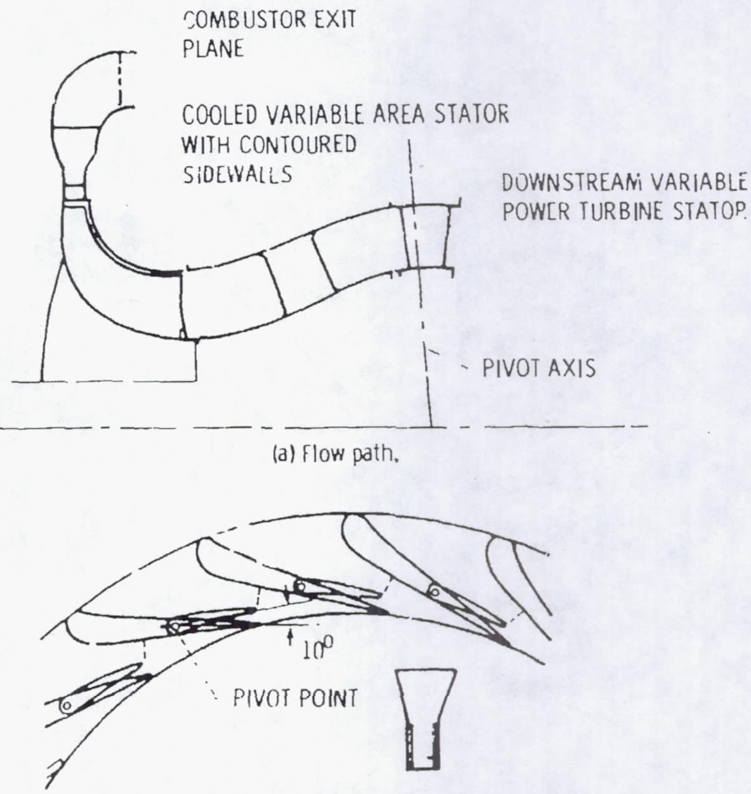


Figure 16. - Stage performance with moveable hub sidewall. $\dot{N}/\sqrt{\theta_{cr}}$ = constant; P_1'/P_4' = constant.



(a) Flow path.

(b) Articulated trailing-edge nozzle area variation concept.

Figure 17. - Cooled, variable-area radial turbine.

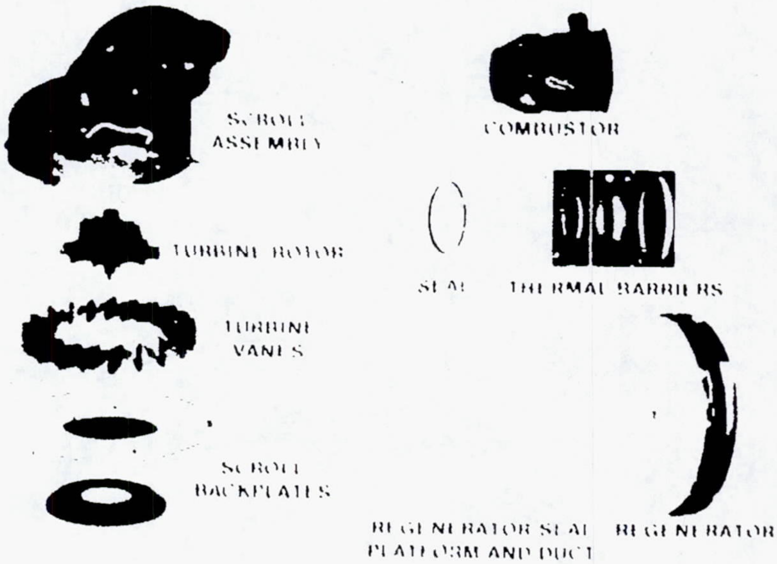


Figure 18. AGT 100 ceramic components.

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Figure 19. - Rotors for improved off design performance

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Figure 20. Turbonetics (MTI) radial steam turbine.

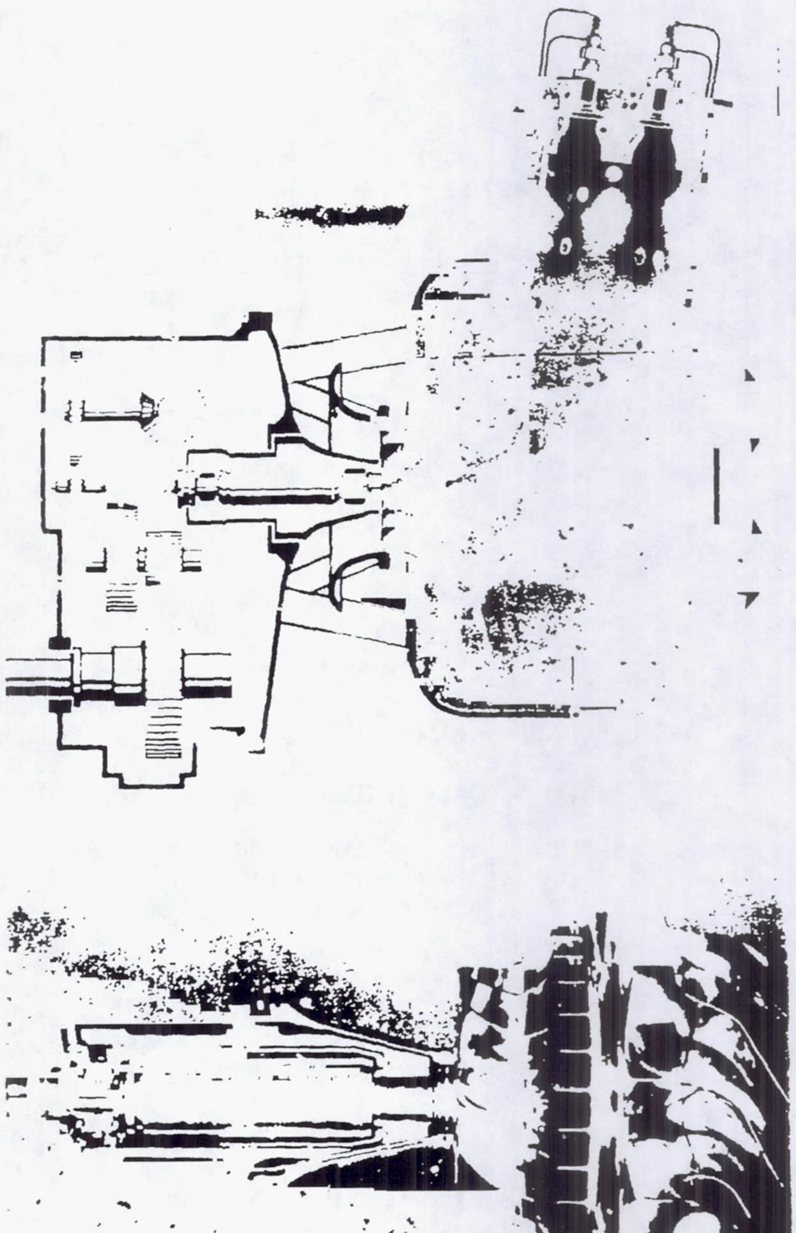

Figure 21. - Kongsberg engine.

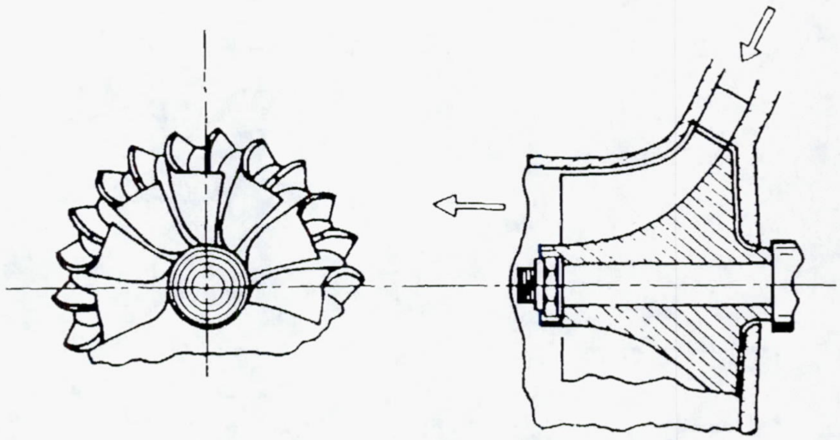

Figure 22. - Mixed flow turbine.

- BASIC BOUNDARY LAYER BEHAVIOR
    - ABILITY TO MODEL TRANSITION AND SEPERATION
    - INFLUENCE OF TURBULENCE, PRESSURE GRADIENT, CURVATURE AND UNSTEADY FLOW
- SECONDARY FLOWS
- TRAILING EDGE LOSSES AND DEVIATION ANGLES
- REYNOLDS NUMBER EFFECTS
- TIP CLEARANCE GEOMETRY/LOSS RELATIONSHIPS
- ACCURACY OF HEAT TRANSFER PREDICTION
- DISK CAVITY LOSSES AND HEAT TRANSFER

Figure 23. Major concerns in turbine technology.

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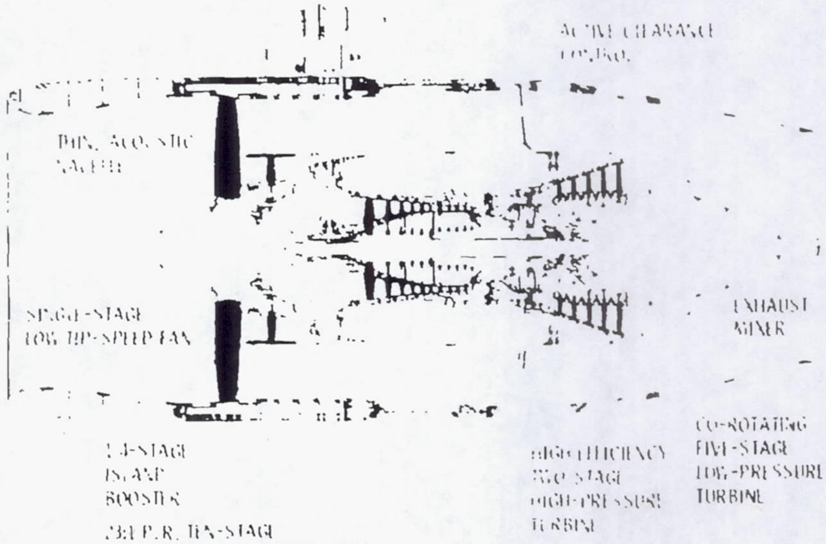
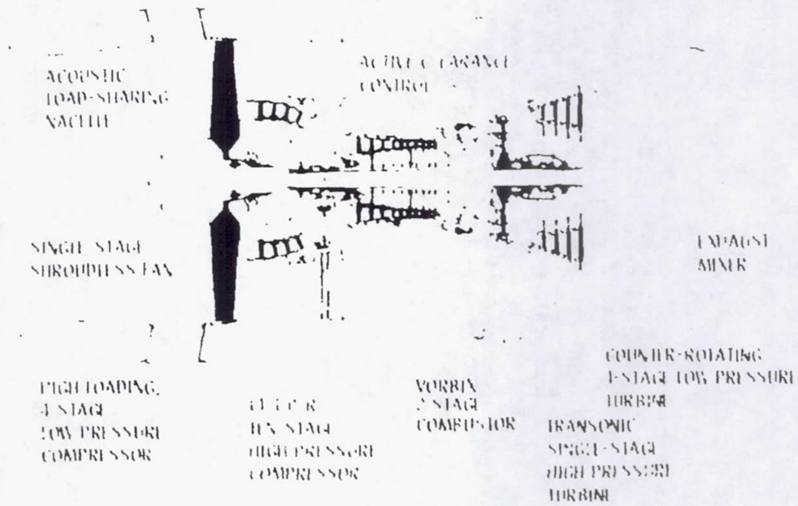


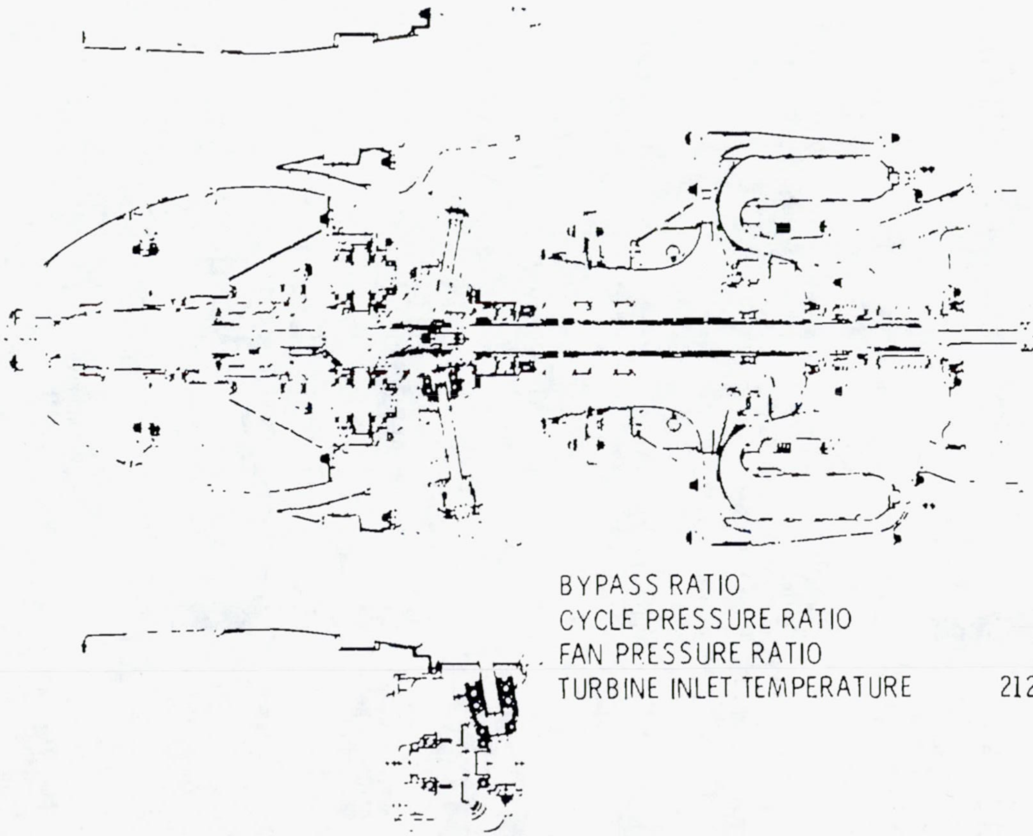
BYPASS RATIO 7.0
CYCLE PRESSURE RATIO 36
FAN PRESSURE RATIO 1.61
TURBINE INLET TEMPERATURE 2440° F

Figure 24. - GE energy efficient engine.



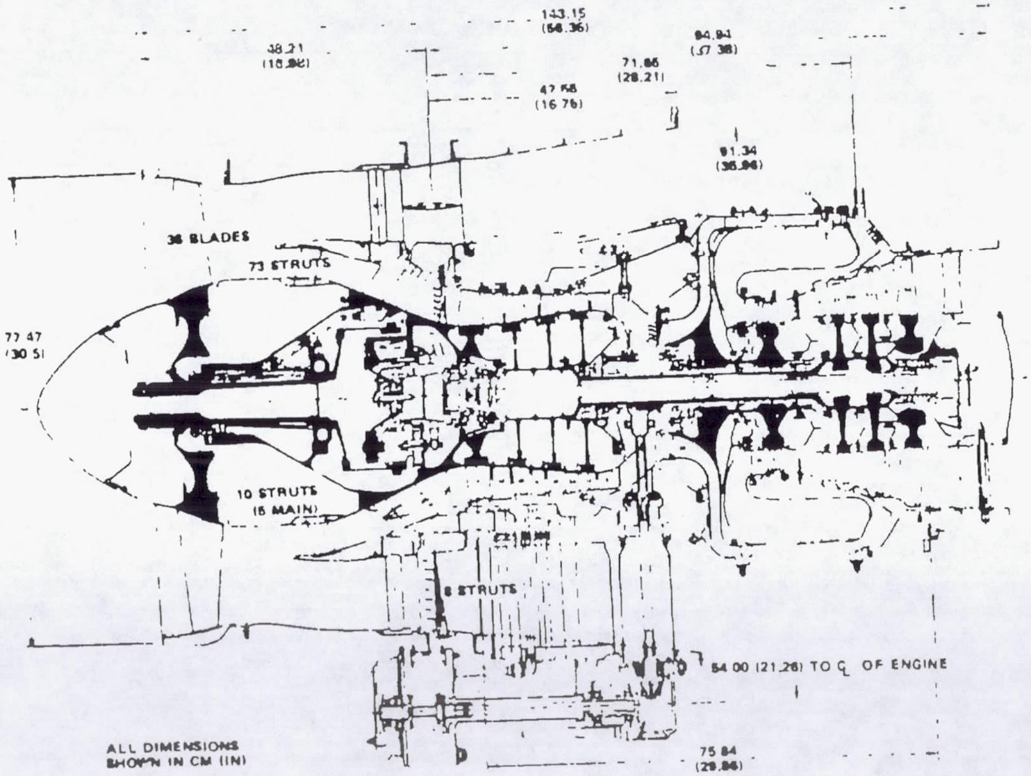
BYPASS RATIO 6.55
CYCLE PRESSURE RATIO 38.6
FAN PRESSURE RATIO 1.74
TURBINE INLET TEMPERATURE 2450° F

Figure 25. - P&W energy efficient engine.



BYPASS RATIO	9.4
CYCLE PRESSURE RATIO	13.7
FAN PRESSURE RATIO	1.36
TURBINE INLET TEMPERATURE	2120^{0} F

Figure 26. - NASA OCGAT cross section-AVCO Lycoming.



BYPASS RATIO	3.7
CYCLE PRESSURE RATIO	17.7
FAN PRESSURE RATIO	1.6
TURBINE INLET TEMPERATURE	1820^0 F

Figure 27. - QCGAT engine cross section-Garrett.

- 3D VISCOUS FLOW CODES
  - LEADING EDGES
  - TRAILING EDGES
  - ROTATION
  - HEAT TRANSFER
  - COOLANT ADDITION
- FLOW AND HEAT TRANSFER CODES FOR LOCAL AREAS
  - COOLING HOLES AND SLOTS
  - COOLANT PASSAGES WITHIN VANES AND BLADES
- FLOW AND HEAT TRANSFER CODES FOR DISK CAVITIES
- WELL DEFINED TURBINE ENVIRONMENT
  - TEMPERATURE AND PRESSURE PROFILES
  - TURBULENCE
  - COOLANT FLOW CONDITIONS AND DISTRIBUTIONS
- REDUCED TIP CLEARANCES AND SENSITIVITY
- INCREASED AN^2

Figure 28. - Long range needs in design technology.

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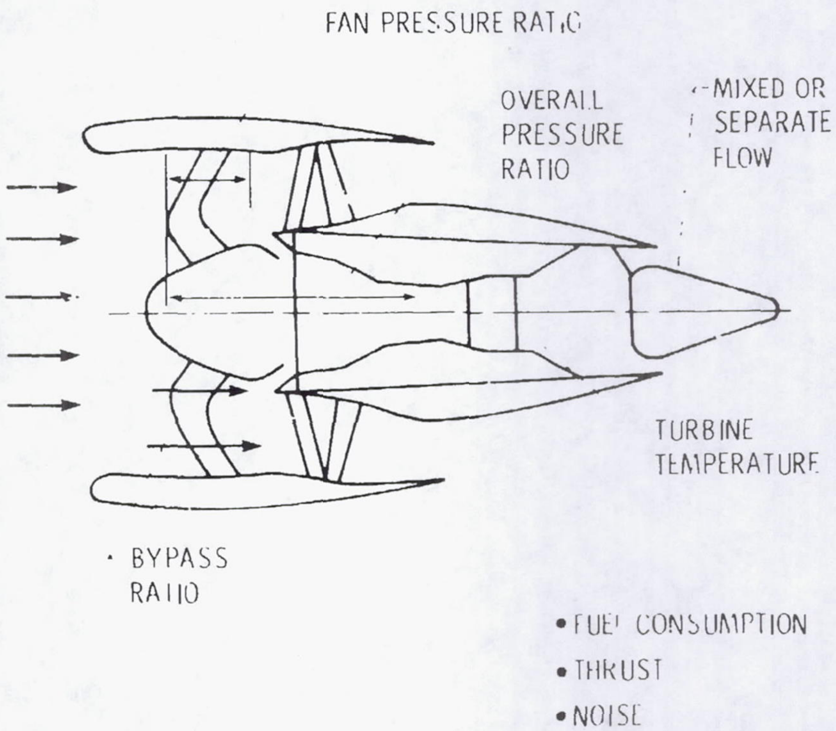

Figure 29. - Elements of cycle studies.

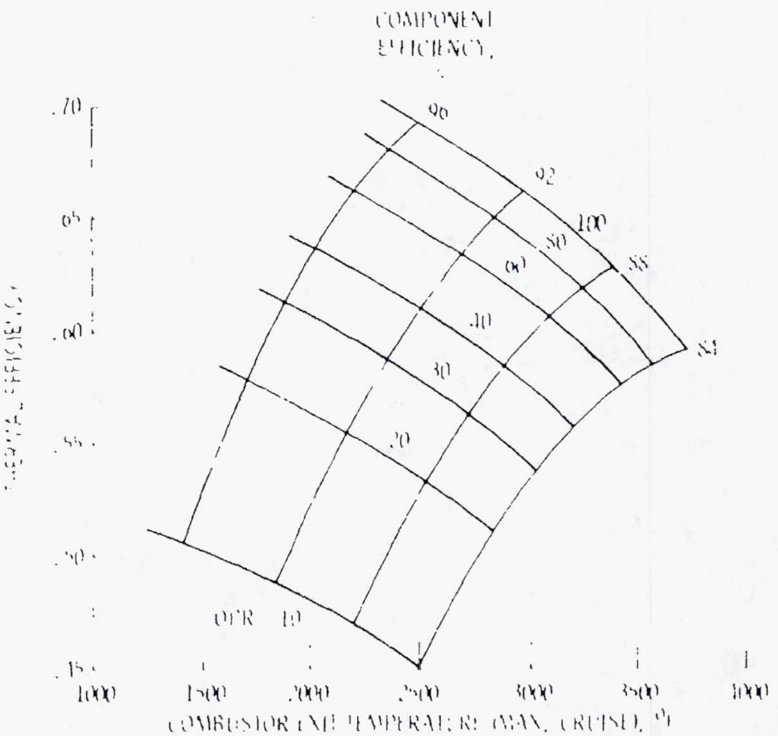

Figure 30. – Turbine temperature for peak thermal efficiency. 35 000 ft; 0.80 Mn; combustor ΔP/P = 0; turbine cooling air = 0.

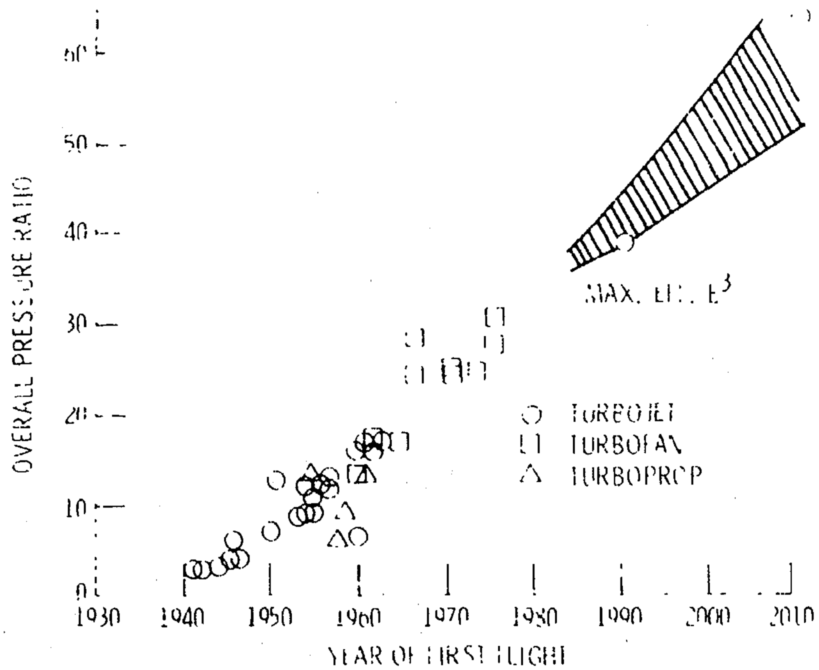

Figure 31. - Progress in pressure ratio.

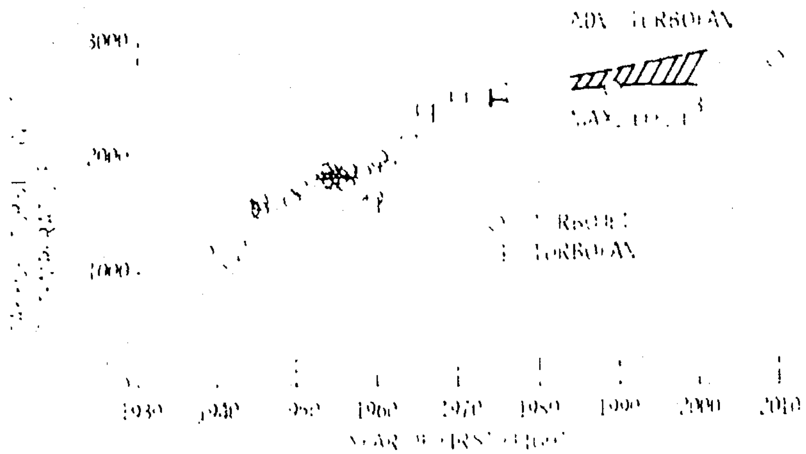

Figure 32. [illegible]

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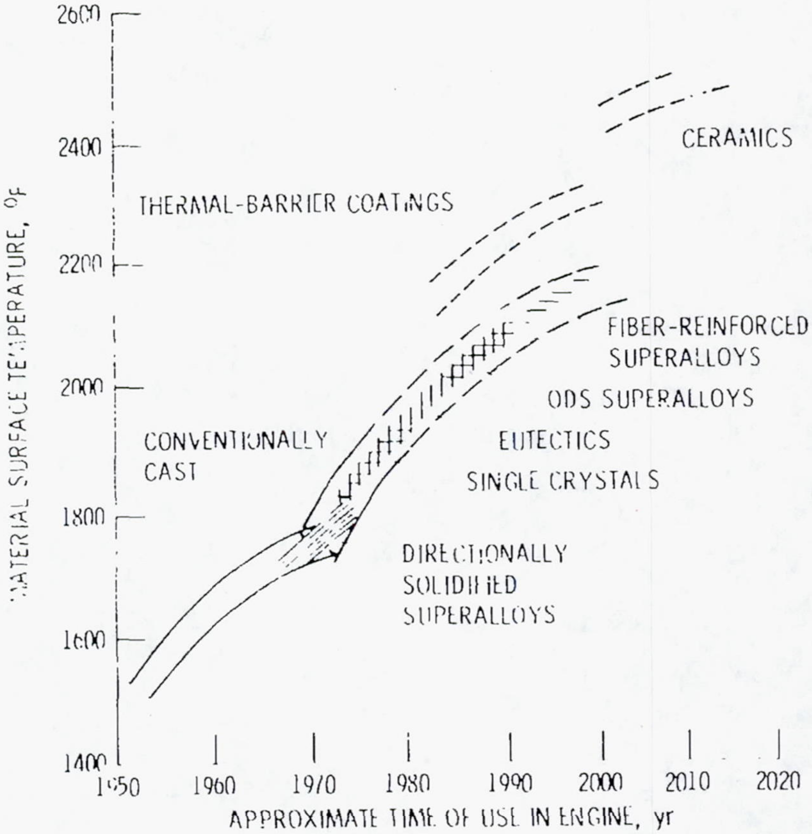

Figure 33. - Temperature capabilities of turbine blade materials.

	LARGE TURBOFAN	LARGE TURBOPROP	SMALL
CORE TURBINE EFFICIENCY	93+	93+	86 TO 91
LOW PRESSURE TURBINE EFFICIENCY	94+	94+	93
COOLANT FLOWS	18 ↓ 6 TO 10%	18 ↓ 6 TO 10%	50% REDUCTION
TURBINE INLET TEMPERATURES, °F	2700 TO 2800 (2400 CRUISE)	2500 TO 2600	2400+
TURBINE INLET PRESSURES, psia (CYCLE PRESSURE RATIOS)	800 TO 900 (55 TO 65)	500 TO 600 (35 TO 90)	250 TO 500 (18 TO 35)

Figure 34. - Turbine performance goals for 1990 to 2000.

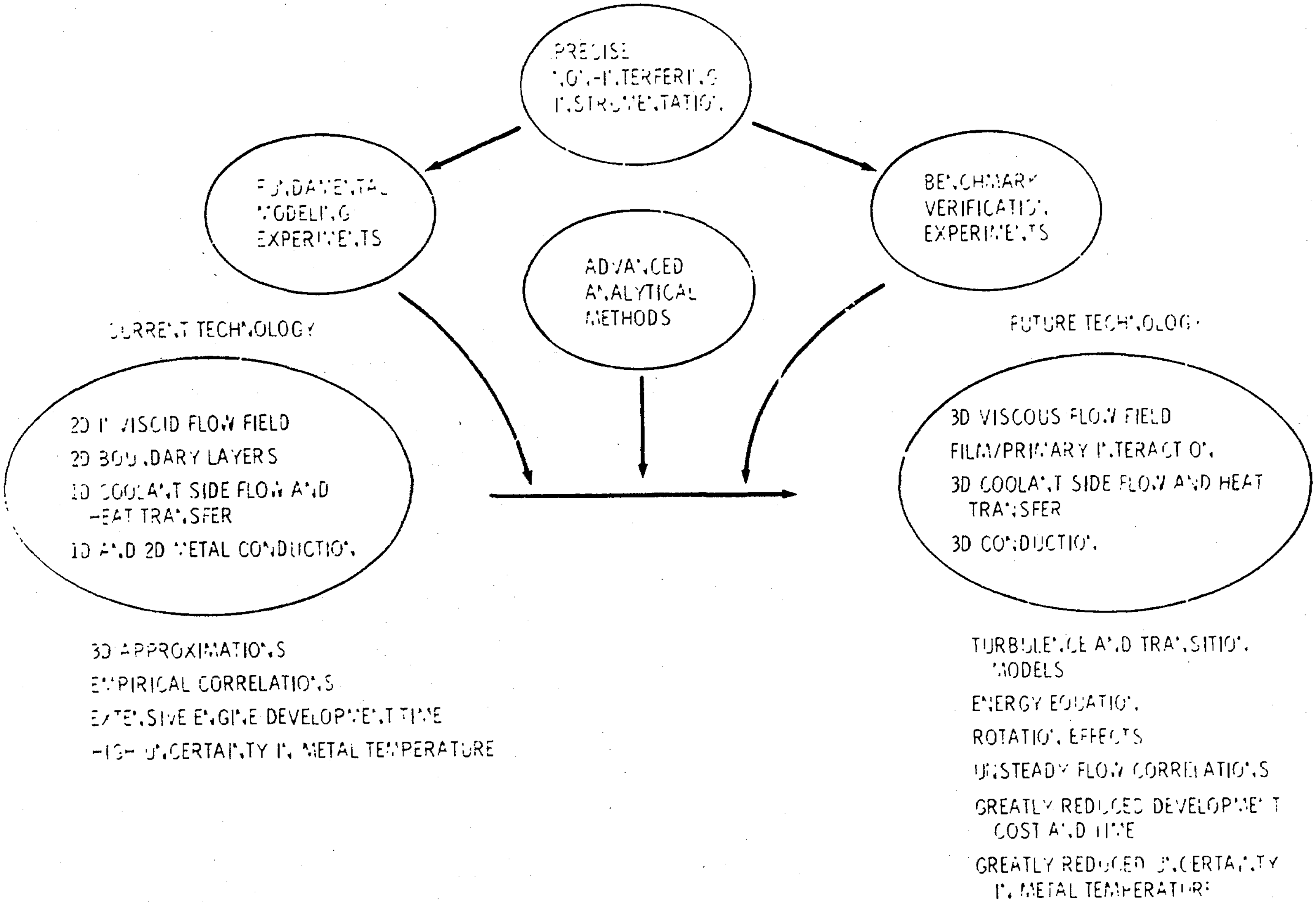

Figure 35. - Technology overview.